Acta Astronautica 145 (2018) 501–514

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

Adaptive extended-state observer-based fault tolerant attitude control for spacecraft with reaction wheels

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ARTICLE INFO

Keywords:
Attitude maneuver
Finite-time convergence
Second-order sliding mode
Fault tolerant control
Adaptive-gain observer

ABSTRACT

This study presents an adaptive second-order sliding control scheme to solve the attitude fault tolerant control problem of spacecraft subject to system uncertainties, external disturbances and reaction wheel faults. A novel fast terminal sliding mode is preliminarily designed to guarantee that finite-time convergence of the attitude errors can be achieved globally. Based on this novel sliding mode, an adaptive second-order observer is then designed to reconstruct the system uncertainties and the actuator faults. One feature of the proposed observer is that the design of the observer does not necessitate any priori information of the upper bounds of the system uncertainties and the actuator faults. In view of the reconstructed information supplied by the designed observer, a second-order sliding mode controller is developed to accomplish attitude maneuvers with great robustness and precise tracking accuracy. Theoretical stability analysis proves that the designed fault tolerant control scheme can achieve finite-time stability of the closed-loop system, even in the presence of reaction wheel faults and system uncertainties. Numerical simulations are also presented to demonstrate the effectiveness and superiority of the proposed control scheme over existing methodologies.

1. Introduction

Nowadays, the attitude control problem of rigid spacecraft has been seriously examined for its extensive applications in space missions. However, due to the nonlinear and highly-coupled attitude dynamics, designing controller to perform large and rapid attitude maneuver is still an open problem. Recently, various control methods, such as adaptive control [1,2], robust control [3,4] and optimal control [5–7], etc., have been carried out to improve the performance of the attitude control system. Among these approaches, sliding mode control (SMC) [8] is an effective strategy due to its great robustness to system uncertainties. The earliest well-known applications of using SMC to accomplish spacecraft attitude control were introduced by Vadali [9] and Dwyer et al. [10]. Hu et al. [11] introduced the SMC to propose a dual-stage attitude maneuver control scheme for a flexible spacecraft. Taking actuator saturation into consideration, Boskovic et al. [12] proposed a globally robust SMC algorithm for the spacecraft attitude stabilization. Cong et al. [13] proposed a backstepping-based SMC for attitude stabilization problem, in which the upper bounds of the system uncertainties and external disturbances are not necessary in the control design.

When solving the attitude control problem via SMC schemes, second-order sliding mode control (SOSMC) is an advanced approach to provide higher tracking accuracy and better robustness [14]. Pukdeboon et al. [14] proposed an SOSMC algorithm for spacecraft attitude maneuver by applying an integral SMC. Applying the geometric homogeneity approach, Tiwari et al. [15] designed a globally robust SOSMC algorithm for rigid spacecraft attitude tracking. In addition, Li et al. [16] developed an asymptotic SOSMC for spacecraft formation flying. Another methodology to improve the robustness and accuracy of the controller is the use of nonlinear observers. To actively compensate for system uncertainties, several authors have incorporated an extended-state observer (ESO) into SMC schemes because an ESO is an effective method to estimate nonlinear dynamics. For instance, Xia et al. [17] proposed an ESO-based SMC attitude control approach, and the designed ESO was applied to estimate the bounded disturbances. Li et al. [18] combined the ESO method with the control Lyapunov function (CLF) approach to accomplish attitude control even in the presence of actuator saturation. Hu et al. [19] proposed a disturbance observer based attitude control scheme in

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https://doi.org/10.1016/j.actaastro.2018.01.021
Received 1 June 2017; Received in revised form 13 July 2017; Accepted 11 January 2018

the presence of disturbance, actuator saturation and misalignment. However, most of the above observer methods need priori information of the upper bounds of the system uncertainties or need to assume a large enough control parameter to guarantee their stability. Nevertheless, these SOSMC and ESO-based SMC controllers ensure a better performance than the common SMC, which in fact involves and demonstrates the potential and benefit of developing an integrated control technique of ESO and SOSMC for spacecraft attitude control. In addition, for spacecraft attitude controller design, terminal sliding mode (TSM) [20] with finite-time convergence is highly preferable and can obtain a superior control performance, i.e. faster convergence rate. Extensive studies of the TSM technique have been carried out to gain finite-time control capability [21–23]. By introducing the TSM surface, Song et al. [24] reported a finite-time attitude controller with inner and outer control loop scheme for rigid spacecraft. Lu et al. [25] investigated a continuous TSM controller to handle the discontinuity problem in TSM controller design. Based on the Lagrange-like modified Rodriguez parameters (MRPs) model, Gao et al. [26] designed an integral SMC to accomplish finite-time attitude control. However, those controllers were proposed based on the assumption that the actuators are fault free.

In addition to the preceding interests in ensuring finite-time convergence, the occurrence of actuator faults is another key issue that should be addressed in spacecraft attitude control, which may lead to performance degradation or instability of the closed-loop system. In order to tackle this issue, fault tolerant control (FTC) is a widely used scheme to improve algorithms' capabilities of maintaining control stability and high performance despite casual actuator failures. Based on the backstepping control method, Hu et al. [27] proposed a fault tolerant control for spacecraft under actuator magnitude deviation and misalignment. To deal with the case that no control torques were applied on either roll or yaw axis, Godard et al. [28] proposed a time-invariant smooth controller to achieve the stabilization of the three-axis attitude. Han et al. [29] designed an adaptive FTC algorithm by using an ideal reference model to identify the actuator faults. Although several finite-time stable FTC schemes were further investigated in Refs. [30,31] by applying TSM approach, only the degradation of actuator's effectiveness was considered. To date, FTC design for spacecraft attitude control with consideration of different kinds of actuator faults and failures is still an open problem.

Motivated by solving the attitude stabilization problem with fault tolerant capability and finite-time convergence, a finite-time stable FTC scheme is designed by integrating an adaptive observer and SOSMC in this paper. The major contributions and differences of this study relative to other papers are the following: (1) To the best of the authors' knowledge, the result presented in this paper is the first attempt to investigate an integrated attitude controller incorporating ESO and SOSMC with finite-time convergence, while the actuator faults are also addressed; (2) In comparison with the aforementioned disturbance observers, the proposed adaptive nonsingular ESO does not require any priori information on the upper bounds of system uncertainties and actuator faults.

The reminder of this paper is organized as described next. In the next section, the mathematical model of the rigid spacecraft, the actuator faults as well as the problem formulation are stated. In Sec.3, an integrated FTC scheme based on ESO and SOSMC is designed for spacecraft attitude maneuvers. The numerical simulations are presented in Sec.4. Finally, the paper ends with conclusion.

Notations: For a given vector $\mathbf{x} = [x_1, x_2, \cdots, x_n]^T \in \mathbf{R}^n$ and $y \in \mathbf{R}$, define $|\mathbf{x}| = [|x_1|, |x_2|, \cdots, |x_n|]^T$, $\mathbf{x}^\beta = [x_1^\beta, x_2^\beta, \cdots, x_n^\beta]^T$ and $\mathrm{sgn}^\beta(\mathbf{x}) = [\mathrm{sgn}^\beta(x_1), \mathrm{sgn}^\beta(x_2), \cdots, \mathrm{sgn}^\beta(x_n)]^T$, where $\beta \in \mathbf{R}$, $\mathrm{sgn}(\cdot)$ is the sign function and $\mathrm{sgn}^\beta(y) = |y|^\beta \mathrm{sgn}(y)$.

2. Mathematical model and preliminaries

2.1. Rigid spacecraft attitude control model

In this subsection, a rigid spacecraft using four reaction wheels is considered for study. Three coordinate frames are defined in this paper, which are the inertial reference frame F_i, the body-fixed frame F_b and the desired attitude frame F_d. Denoting the four reaction wheels by RW_1, RW_2, RW_3 and RW_4, it is assumed that reaction wheels RW_1, RW_2 and RW_3 are installed with spin axes parallel to the spacecraft body axes, and RW_4 is installed with spin axis pointing in a specific direction. The attitude kinematics and dynamics are described by Ref. [32].

$$\begin{aligned} \dot{q}_0 &= -\frac{1}{2}\bar{\mathbf{q}}^T\boldsymbol{\omega} \\ \dot{\bar{q}} &= \frac{1}{2}\left(q_0\mathbf{I}_3 + \bar{\mathbf{q}}^\times\right)\boldsymbol{\omega} \end{aligned} \tag{1}$$

$$\mathbf{J}_s\dot{\boldsymbol{\omega}} = -\boldsymbol{\omega}^\times(\mathbf{J}_s\boldsymbol{\omega} + \mathbf{D}\mathbf{J}_w\boldsymbol{\Omega}) + \mathbf{D}\mathbf{u} + \boldsymbol{\tau}_d \tag{2}$$

where $\mathbf{q} = [q_0, \bar{\mathbf{q}}^T]^T \in \mathbf{R}^4$ denotes the unit quaternion describing the attitude of F_b with respect to F_i; $q_0 \in \mathbf{R}$ and $\bar{\mathbf{q}} \in \mathbf{R}^3$ are the scalar component and the vector part of $\mathbf{q}$, respectively; $\boldsymbol{\omega} \in \mathbf{R}^3$ represents the body angular velocity in F_b relative to F_i; $\mathbf{I}_n$ represents the $n \times n$ identity matrix; for a vector $\mathbf{a} \in \mathbf{R}^3$, $\mathbf{a}^\times \in \mathbf{R}^{3\times3}$ is a corresponding skew-symmetric matrix such that $\mathbf{a}^\times\mathbf{b}$ yields the skew-symmetrical matrix between $\mathbf{a}$ and $\mathbf{b}$ for any $\mathbf{b} \in \mathbf{R}^3$; $\mathbf{J}_s \in \mathbf{R}^{3\times3}$ denotes the spacecraft inertia matrix with nominal part $\mathbf{J}$ and uncertain component $\Delta\mathbf{J}$ such that $\mathbf{J}_s = \mathbf{J} + \Delta\mathbf{J}$; $\mathbf{u} \in \mathbf{R}^4$ represents the control torque generated by the reaction wheels; $\mathbf{D} \in \mathbf{R}^{3\times4}$ represents the reaction wheels orientation matrix; $\boldsymbol{\tau}_d \in \mathbf{R}^3$ denotes the bounded but unknown external disturbances. Moreover, $\mathbf{J}_w \in \mathbf{R}^{4\times4}$ is a diagonal matrix containing the reaction wheel inertias about their respective spin axes along the matrix diagonal, and $\boldsymbol{\Omega} \in \mathbf{R}^{4\times1}$ is a vector containing the reaction wheel spin rates. Specifically, $\mathbf{u}$ provided by the reaction wheels is given by $\mathbf{u} = -\mathbf{J}_w\dot{\boldsymbol{\Omega}}$.

To address the attitude maneuver issue, an error quaternion $\mathbf{q}_e = \begin{bmatrix} q_{e0} & \bar{\mathbf{q}}_e^T \end{bmatrix}^T \in \mathbf{R}^4$ is defined as the relative attitude error between F_b and F_d with desired quaternion $\mathbf{q}_d \in \mathbf{R}^4$. Then one can obtain $\mathbf{q_e} = \mathbf{q}^{-1} \otimes \mathbf{q}_d$, where '$\otimes$' represents the quaternion multiplication. The relative angular velocity $\boldsymbol{\omega}_e \in \mathbf{R}^3$ in F_b relative to F_d is defined as $\boldsymbol{\omega}_e = \boldsymbol{\omega} - \mathbf{C}\boldsymbol{\omega}_d$, where $\boldsymbol{\omega}_d \in \mathbf{R}^3$ denotes the desired angular velocity expressed in F_d and $\mathbf{C}$ represents the corresponding rotation matrix between F_b and F_d, which is given by $\mathbf{C} = (q_{e0}^2 - \bar{\mathbf{q}}_e^T\bar{\mathbf{q}}_e)\mathbf{I}_3 + 2\bar{\mathbf{q}}_e\bar{\mathbf{q}}_e^T - 2q_{e0}\bar{\mathbf{q}}_e^\times$.

Hence, the relative error attitude kinematics and dynamics can be described by Refs. [33,34].

$$\begin{aligned} \dot{q}_{e0} &= -\frac{1}{2}\bar{\mathbf{q}}_e^T\boldsymbol{\omega}_e \\ \dot{\bar{q}}_e &= \frac{1}{2}\mathbf{M}(\bar{\mathbf{q}}_e)\boldsymbol{\omega}_e \end{aligned} \tag{3}$$

$$\mathbf{J}\dot{\boldsymbol{\omega}}_e = -\boldsymbol{\omega}^\times(\mathbf{J}\boldsymbol{\omega} + \mathbf{D}\mathbf{J}_w\boldsymbol{\Omega}) - \mathbf{J}\left(\mathbf{C}\dot{\boldsymbol{\omega}}_d - \boldsymbol{\omega}_e^\times\mathbf{C}\boldsymbol{\omega}_d\right) + \mathbf{D}\mathbf{u} + \boldsymbol{\tau}_d + \Delta\boldsymbol{\tau} \tag{4}$$

where $\mathbf{M}(\bar{\mathbf{q}}_e) = q_{e0}\mathbf{I}_3 + \bar{\mathbf{q}}_e^\times$; $\Delta\boldsymbol{\tau}$ represents the disturbance torque caused by inertia uncertainties, which is defined as $\Delta\boldsymbol{\tau} = -\Delta\mathbf{J}(\dot{\boldsymbol{\omega}}_e + \mathbf{C}\dot{\boldsymbol{\omega}}_d - \boldsymbol{\omega}_e^\times\mathbf{C}\boldsymbol{\omega}_d) - \boldsymbol{\omega}^\times\Delta\mathbf{J}\boldsymbol{\omega}$.

2.2. Actuator faults or failure model

As discussed in Refs. [35,36], reaction wheel is sensitive to four kinds of faults or failures such as *Decreased reaction torque* (F1); *Increased bias torque* (F2); *Continuous generation of reaction torque* (F3); *Failure to respond to control signals* (F4). For each wheel, those faults or failures are modelled as

$$u_i = e_i(t)u_{ci} + \bar{u}_{ci}, \qquad i = 1, 2, 3, 4 \tag{5}$$

where u_{ci} represents the control torque produced by the designed control algorithm for the ith wheel; e_i denotes the effectiveness of the ith wheel satisfying $0 \le e_i \le 1$; $\bar{u}_{ci}$ represents the uncertain input part for the ith